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JOINT INSTITUTE FOR AERONAUTICS AND ACOUSTICS
STANFORD UNIVERSITY

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NASA GRANT NCC 2-55 #29

THE RESEARCH AND TRAINING ACTIVITIES FOR THE JOINT INSTITUTE FOR AERONAUTICS AND ACOUSTICS

Proposal submitted to the
NASA Ames Research Center
Moffett Field, CA 94035

For a period of One Year
October 1, 1996 to September 30, 1997

by the

Department of Aeronautics and Astronautics
Stanford University
Stanford, California 94305

Principal Investigator
Professor Brian Cantwell

September 1996

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ABSTRACT

This proposal requests continued support for the program of activities to be undertaken by the Ames-Stanford Joint Institute for Aeronautics and Acoustics during the one-year period October 1, 1996 to September 30, 1997. The emphasis in this program is on training and research in experimental and computational methods with application to aerodynamics, acoustics and the important interactions between them. The program comprises activities in active flow control, Large Eddy Simulation of jet noise, flap aerodynamics and acoustics, high lift modeling studies and luminescent paint applications. During the proposed period there will be a continued emphasis on the interaction between NASA Ames, Stanford University and Industry, particularly in connection with the noise and high lift activities.

The program will be conducted within the general framework of the Memorandum of Understanding (1976) establishing the Institute, as updated in 1993. As outlined in the agreement, the purposes of the institute include the following:

- To conduct basic and applied research.
- To promote joint endeavors between Center scientists and those in the academic community.
- To provide training to graduate students in specialized areas of aeronautics and acoustics through participation in the research programs of the Institute.
- To provide opportunities for Post-Doctoral Fellows to collaborate in research programs of the Institute.
- To disseminate information about important aeronautical topics and to enable scientists and engineers of the Center to stay abreast of new advances through symposia, seminars and publications.

The program described above is designed to address future needs of NASA Ames and has been the basis of discussion among Professors B. Cantwell, I. Kroo, S. Lele and S. Rock from the Stanford faculty and several members from NASA Ames including Dr. S. Davis, Dr. Dochan Kwak, Dr. S. Smith, Dr. J. Ross and Dr. L. Olsen. Coordination of this activity at Ames is the responsibility of the Institute Associate Director for Center Affairs, Dr. C. A. Smith.

1. Joint Institute Program Overview

1.1 Introduction

Experimental and computational aerodynamics have for many years played an important role in the basic and applied research programs of Ames Research Center and in the research and training activities of Stanford University. Recently, computational tools have been brought to bear on the difficult problem of flow generated noise. The coordinated use of a combination of experimental and computational tools has long been recognized as an essential part of a comprehensive approach to improving our fundamental understanding of complex flow phenomena. Developments in computational capabilities, in flow visualization, in measurement and in new kinds of wind-tunnel instrumentation will constitute a major step forward in the ability of scientists and engineers to advance the state of the art in aerodynamic design technology.

It is therefore the general character of the proposed program that it involves both experiment and computation and that these are used in complimentary ways. This approach can be undertaken only if highly qualified personnel and good research facilities are available. In this regard the blending of resources from Stanford and Ames is an important ingredient and was one of the motivating reasons behind the establishment of the Ames-Stanford Joint Institute.

In the experimental parts of the program described below, smaller scale investigations undertaken at Stanford are coordinated with both computations and experiments carried out in the more powerful facilities at Ames Research Center.

1.2 Research Project Summaries

The research directions summarized here, and further elaborated in the Program Description, are the result of several discussions with research management and staff at Ames Research Center. The activities are consistent with the emphasis on acoustics and high-lift in current NASA programs.

Project 1 - Active flow control

This is a continuing program in the use of active flow control as a means of regulating aircraft attitude at high angles of attack. The combined roll and yaw control of a generic thin delta wing aircraft using fore-body tangential blowing is being investigated. Techniques for developing nonlinear optimum control laws are being established using results obtained from a unique free-to-roll, free-to-yaw support system. Wind tunnel data and numerical simulations are being used to provide the aerodynamic information necessary in the formulation of control laws for this configuration.

Project 2 - LES as a tool for studying jet aero-acoustics

New subsonic and supersonic aircraft are required to meet increasingly more stringent environmental noise regulations. Current design/analysis tools for estimating the noise generated by an aircraft configuration rely strongly on

empirical formulations. With recent advances in computational technology it seems possible that important components of aircraft noise could be predicted by a theoretical approach. Since noise is generated by unsteady flow it becomes necessary to accurately predict the unsteady flow. The proposed research seeks to evaluate and develop Large Eddy Simulation (LES) as a computational technology for predicting jet-noise.

Project 3 - Research on a lifting wing-flap configuration

The adoption of lower tolerance international, national and local noise regulations and the advent of large, high lift commercial aircraft has led to a renewed interest in noise generation by airframe components. Recent studies of airframe noise have identified the wing and flap trailing edge as well as the flap side-edge as areas of elevated noise generation. In this project the fluid dynamic processes associated with these two classes of noise sources are being investigated. A NACA 63-215 Mod B airfoil section has been used by NASA Ames investigators for high Reynolds number experiments in the Ames 7x10 tunnel. These experiments included noise studies carried out by Boeing and Ames investigators using Boeing-developed phased array instrumentation. This same geometry is also being studied in CFD computations by Ames and Stanford investigators and in small scale experiments at Stanford University. The Stanford experiments emphasize: mapping the mean flow and turbulence quantities in the near wake of the flap side-edge; making unsteady pressure measurements over the flap and other sections of interest; and performing visualization and measurement of unsteady aspects of the flow which can not be easily studied in either the computations or the 7x10 experiments.

Project 4 - Luminescent paint for aerodynamic measurement

This project employs luminescent (pressure sensitive) paint to measure the spatial pressure distribution on a wind tunnel model. The emphasis is on extending the technique to low speeds and unsteady flows. Pressure sensitive paints are based on a class of chemicals known as porforins and make use of a surface reaction which, under illumination with ultraviolet light, causes the scattered light intensity to be proportional to the partial pressure of oxygen at the painted surface. The variation in scattered intensity can recorded with a video camera and used to infer surface pressure over an extended area. With further development these paints, along with similar systems capable of measuring wall shear stress, promise to revolutionize wind tunnel testing techniques. In particular, the high cost of pressure instrumentation for wind tunnel models can be greatly reduced. The surface reaction is fast and in principle it should be possible to measure a time varying pressure. Initial, proof-of-concept, experiments have been completed at Stanford using a jet impinging on a plate to examine the time dependent response of the paint. Low speed measurements of the mean pressure distribution on a delta wing have been made in the Stanford Subsonic Windtunnel.

Project 5 - Prediction of wing maximum lift for preliminary design

The high lift characteristics of wings have important effects on aircraft noise, cost, and performance. The proposed research is aimed at improved understanding of the high lift flow regime for high aspect ratio wings in the context of preliminary analysis and design. Computational models are being developed and used to examine the important inviscid and viscous phenomena that effect wing maximum lift, as well as the importance of three-dimensionality on the flow field. The ultimate goal is to develop an analysis routine that accurately predicts wing maximum lift with the speed, accuracy, and sensitivity necessary for use in multidisciplinary optimization design codes.

Project 6 - New methods for aero-acoustic predictions at low speed

The prediction and control (reduction) of aerodynamically generated noise is important to the design of quieter future aircraft. Airframe noise associated with high-lift devices in a landing configuration needs to be reduced in designing new subsonic airplanes. Wind noise impact of high-speed trains is also becoming increasingly important. High-speed trains travelling between city centers must pass through areas busy with urban communities and concerns about the community noise impact are rising. The common factor in these two examples is that the flow processes responsible for noise-generation occur at essentially incompressible conditions. Developing general computation based prediction methods for aerodynamically generated sound in the low Mach number regime is the focus of the proposed project.

1.3 Institutional Support

Institutional support involves administrative, secretarial and technical salaries, travel, university equipment and services including communication, expendable supplies, computer services, engineering services, etc., and capital equipment. This support provides all of the basic services necessary for continuing operations of the Institute including its small-scale experimental and computational facilities, instrumentation and equipment and thereby supports all of the research and training activities summarized previously.

1.4 Training activities

The training role of the Institute is accomplished through 6 units of coursework in acoustics offered by the Aero/Astro department including AA 201A (Fundamentals of Acoustics) and AA 201B (Topics in Aeroacoustics).

1.5 Research Participation

The research programs summarized in Item 1.2 above will be undertaken by Stanford faculty, staff and graduate students within the Department of Aeronautics and Astronautics with the involvement of 4 Professors, 2 Research Associates and 5 Ph.D. students. This group has experience in Aerodynamics and Acoustics and is familiar with NASA's wind tunnel and computational facilities. The strong collaboration between Stanford and Ames researchers which

has been the hallmark of Joint Institute research in the past will be continued and enhanced in the coming year. The program activity at Ames will be coordinated by the Institute Associate Director for Center Affairs, Dr. C.A. Smith.

2. Detailed Program Description

The research program proposed for the year, October 1, 1996 to September 30, 1997 is described in detail below. It has been discussed with the cognizant personnel at Ames and agreement has been reached on the general scope of the programs.

2.1 Project 1 - Active flow control

Research participants: Prof. S. Rock, graduate student

Ames Technical Contact: Dr. C.A. Smith

2.1.1 Introduction

Controlled flight at high angles of attack provides increased maneuverability for fighter aircraft and increased lift during take off and landing. At these flight regimes, flow separation and vortex breakdown decrease the efficiency of conventional control surfaces when they are most needed to combat the onset of asymmetric flow. As a result of the inefficiency of the conventional control surfaces at these high angles of attack, alternate means to supplement the vehicle flight control are necessary. This research investigates the augmentation of aircraft flight control systems by the injection of a thin sheet of air tangentially to the fore-body of the vehicle. The method known as Fore-body Tangential Blowing (FTB) is proposed as an effective means of altering the flow over the fore-body of the vehicle (Ref. [1.1] and Ref. [1.2]). By using this method, the flow asymmetries are changed and consequently the aerodynamic loads are modified (Ref. [1.1] to Ref. [1.6]).

Static and dynamic experiments performed at the Department of Aeronautics and Astronautics at Stanford University under the NASA-JIAA program have shown that significant side force, roll and yaw moments as well as normal force and pitching moment (Ref. [1.3], Ref. [1.4] to Ref. [1.7]) can be generated using a small amount of blowing. This is important given that the implementation on a real aircraft would provide a limited amount of air. It has also been demonstrated that FTB could successfully be used to suppress wing rock and to roll the model to a desired bank angle. In addition, it was shown that asymmetric FTB could provide the necessary aerodynamic force and moments for regulating aircraft yaw and roll and slewing to non-zero roll and yaw angles (Ref. [1.4], Ref. [1.5]).

During this past year, dynamic experiments have been conducted in the Stanford Low Speed Wind Tunnel using a wind tunnel model which is allowed two degrees of freedom: roll; and yaw (Ref. [1.4], Ref. [1.5]). This system provides a good approximation of the characteristics of the lateral-directional dynamics of an aircraft. The wind tunnel model currently consists of a cone-cylinder fuse-

lage and a sharp leading-edge delta wing. While a vertical tail can be added, no movable surfaces are installed for control purposes. The model is equipped with fore-body side slots through which blowing is applied. The amount of air injected is controlled by a closed loop control system employing specially designed servo valves and flow meters (Ref. [1.4] to Ref. [1.6]). A view of the wind tunnel test section with the unique two degrees of freedom apparatus and the wind tunnel model is shown in Figure 1.1.

Using a fundamental understanding of the underlying physics and experimental data measured using the wind tunnel model, an unsteady aerodynamic model was developed and validated (Ref. [1.4] and Ref. [1.5]). The model uses a small number of parameters to describe the physics of the flowfield and the impact of blowing, prior to linearization. The model provides the information necessary for real-time regulation about small angles with sufficient accuracy and thus forms the basis for demonstrating FTB. It will not support real-time control for commanding large angle motion because of its limited scope and does not explicitly describe important physical phenomena such as vortex breakdown. Further understanding of these phenomena would be required before FTB could be used on real aircraft under operational conditions.

2.1.2 Research Objectives

Current research focuses on further developing our understanding of the fundamental physics of the interactions between an aircraft's attitude and motion, the aerodynamic forces and moments generated by the vortical flowfield and the impact of adding FTB as well as developing new nonlinear control approaches to fully exploit the benefits of FTB. An investigation into modifying the geometry of the model's fore-body is also being undertaken. Recent experimental and computational studies have demonstrated that the vortical structure generated by a chined fore-body will enhance lateral stability at high angles of attack without the aid of blowing (Ref. [1.8] to Ref. [1.10]). Other studies have shown the promising potential of blowing off a chined body at low to medium angles of attack (Ref. [1.11] to Ref. [1.12]). However, there are still-unresolved questions regarding the efficiency of combining FTB with a chined nose for lateral control at high angles of attack.

2.1.3 Research Program

Experimental investigations have been conducted with a wind tunnel model with yaw and roll degrees of freedom. Static and dynamic measurements of the aerodynamics have been used to characterize the natural behavior of the system and the effect of blowing. A nonlinear mathematical extension of the current model of the system is being generated for use in the synthesis of control laws to provide the capability to command large roll and yaw angles, ϕ and γ respectively. Past work (Ref. [1.3] to Ref. [1.5]) has shown that blowing has a significant impact on the structure of the vortical flowfield, and can be harnessed to provide the control authority required to regulate aircraft roll and yaw and slew the aircraft to non-zero yaw and roll and angles. Furthermore,

FTB is a nonlinear actuator is not represented well as an incremental control device because the aircraft stability derivatives depend nonlinearly on the level of blowing. The models generated by Wong and Pedreiro lump the effects of FTB, using a small number of parameters to determine the dynamics of the vortical flowfield, prior to linearization. The existing models provide the necessary information required for real-time regulation about small angles and thus the basis for demonstrating that FTB augments the vehicle's conventional control systems. However, they are not capable of supporting real-time control for commanding large angle motion because of the corresponding increase in model complexity and detail required to support such a task. The new model would be required to present a more detailed picture of the aerodynamics to help predict the behavior of the moments and loads on the aircraft model as it undergoes large angle motion. Using FTB in real aircraft under realistic conditions requires further understanding of the underlying physics of such phenomena as vortex breakdown. This understanding can only be achieved through the development of a nonlinear model. Such a model would be accurate and detailed enough to provide reliable predictions for vehicle design and optimization.

Using the apparatus that was previously designed and built under the NASA-JIAA program, dynamic and static experiments were carried out to extend the current understanding of the aerodynamics of the phenomena and the use of blowing to control the roll-yaw motion of an aircraft at high angle of attack. The following results were achieved in the 1995-96 period.

- Experimental demonstration of roll-yaw control at high angle of attack using only FTB in the wind tunnel.
- Development and validation of an unsteady aerodynamic model that includes the effect of blowing and is suitable for controls.
- Development of an approach that uses fore-body tangential blowing as the only actuator to control the roll-yaw motion of the wind tunnel model.

In addition, an investigation of possible methods with which to extend the unsteady aerodynamic model was completed during this past period. This proposed extension to the model would allow for the next logical step in developing fore-body tangential blowing as a control actuator, i.e.; commanding large roll and yaw angles. A discussion of the proposed approach using nonlinear indicial response (NIR) methods will be presented below.

Method of Approach

The stabilization of the system with two degrees of freedom has been demonstrated. The next logical step is to investigate the possibility of commanding large roll and yaw angles. This problem is of utmost interest because it applies directly to the ability to point the vehicle in a desired direction. The character-

istics of the system vary for large roll and yaw angles making the control problem extremely difficult (Figure 1.2) Thus, the challenge is to model these variations so that control logic can be effectively applied.

The existing model lumps the effects of FTB, using a small number of parameters to determine the dynamics of the vortical flowfield, prior to linearization. Although it does provide the necessary information required for real-time control with sufficient accuracy and thus the basis for demonstrating that FTB augments the vehicle's conventional control systems, it will not explicitly describe important physical phenomena that are strong contributors to the nonlinear behavior of the aerodynamic loads and moments at non-zero yaw and roll angles. Before the use of FTB in real aircraft under realistic conditions can be contemplated, further understanding of the underlying physics of the phenomena is required. This understanding can only be achieved through the development of a nonlinear model that directly addresses such important issues. Such a model would be accurate and detailed enough to provide reliable predictions for vehicle design and optimization.

Progress has been made in defining an approach to extend the current aerodynamic model, in order to examine the functions of roll and yaw rates and their angular accelerations with respect to large angle commands. The most promising approach will be attempted first, and calls for the use of indicial response methods to augment present techniques. These methods allow for a more accurate understanding of the unsteady nature of the flowfield, which varies for large roll and yaw angles and includes bifurcations and hysteresis (Ref. [1.13] and Ref. [1.14]).

There are other options for extending the modeling and should an NIR-based model prove to be unworkable, they will applied to solve the task of modeling the coupling between aircraft attitude and motion, the induced aerodynamic loads and the effect of FTB. They apply analytical and semi-empirical methods to model the nonlinear aerodynamics of wings and bodies. The methods that have been considered include nonlinear lifting surface theory and leading edge vortex models that have been modified for the existence of a fore-body and FTB.

A good understanding of indicial response methods, has been achieved. The steps required to extend the existing theory are currently being mapped out and will provide a rigorous approach to the task of modeling the unsteady and nonlinear interaction between the vortical flowfield and tangential blowing.

An indicial response to a step change in a motion is formed by taking the difference between the responses to two motions whose histories differ only by the step imposed on the second motion (Ref. [1.15]). A review of the work done to date would indicate that linear indicial theory is valid away from bifurcations associated with changes in flow topology, provided that the perturbation displacements are small. In general, the linear indicial method cannot handle multiple solutions such as bifurcations, jumps, and other nonlinear phenomena which typically occur in high angle-of-attack vortical flows. Another difficulty

is that the flowfields generated are characterized by a significant amount of hysteresis. This violates the assumptions which underlie the use of linear indicial theory because the linear indicial functions do not contain any memory effects, and depend only on the instantaneous state of the system (Ref. [1.16]).

Nonlinear indicial response (NIR) theory offers a viable alternative which can fulfill the need for the efficient and accurate modeling of a nonlinear plant. NIR functions can be defined in terms of two motions such as those shown in Figure 1.3, which, for example, might describe an aircraft in pure roll. Both share the motion history $\phi(\xi)$, which serves to initialize the flow field at step onset. In the first (denoted as ϕ_1), the motion following the onset of the step at $t = \tau$ is held constant for all time thereafter at $\phi(\tau)$. The second motion, ϕ_2 involves a discrete jump to $\phi(\tau) + \Delta\phi$ at $t = \tau$, and is held constant thereafter. It should be noted that, while finite jumps are not physically possible, the model is both physically and mathematically proper for infinitesimal step heights. When the difference is taken in the linear case, the influence of the identical past cancels perfectly. However, in the absence of linearity, a remnant of the past must exist in the behavior of the indicial response because exact cancellation of the identical past cannot be assumed. Thus the indicial response must be a functional. A functional $F(y(\xi))$ is defined to be an operation that assigns a value to each function $y(\xi)$ of the set of functions (all of which are defined in some interval) for which the functional is defined, much like a function f(x) assigns a value to each x for which it is defined (Ref. [1.19]). Suitable representations can be found in a similar manner for the effects of aircraft motion on other aerodynamic loads and moments. This methodology will account for hysteresis and other nonlinearities exhibited by the plant.

Following the notation of Tobak et al (Ref. [1.13], Ref. [1.20] and Ref. [1.21]) and Jenkins (Ref. [1.15]), the formal definition of the NIR for the rolling moment due to aerodynamic roll angle is given by:

$$C_{l\phi}[\phi(\xi);t,\tau] = \lim_{\Delta\phi \to 0} \frac{C_l(\phi_2(t)) - C_l(\phi_1(t))}{\Delta\phi} \tag{2.1}$$

On the left hand side, the functional dependence on motion history is denoted by the entry to the left of the semi-colon, $\phi(\xi)$. The symbol ξ indicates that the function ϕ is defined over the range $-\infty < t \leq \tau$ and is held constant thereafter.

Equation (2.1) defines the Fréchet derivative of the functional $C_l[\phi_1(t)]$. Tobak et al (Ref. [1.21]) have suggested that bifurcations of physically realizable (asymptotically stable to small perturbations) steady-state “solutions” corresponding to ϕ_1 are signaled by loss of Fréchet differentiability. Such occurrences are of considerable interest to the study of hysteresis effects (Ref. [1.14]).

The response to an arbitrary motion input, just as in the linear case, is obtained by superposition of responses to successive steps. The essential difference is that the response of each step depends on its motion history in the nonlinear case. Superposition is valid if the responses (after step onset) can be obtained from a linear perturbation of the Navier-Stokes equations. This can be done if the equilibrium flow at step onset is asymptotically stable to small perturbations, i.e., no discontinuities or bifurcations may exist in the static response at $\phi(\tau)$. Where equilibrium-flow stability is lost (a critical state marked by a change in flow topology) the Navier-Stokes equations are required and superposition is not valid. In this case, the superposition integral must be split to avoid the singularity and a fully nonlinear time-dependent term added to account for the transition to a new flow state. This is shown below for a critical state encountered at $t = \tau_c$. The resulting rolling moment response to an arbitrary roll angle input is:

$$C_l(t) = C_l(t, \phi(0)) + \int_0^{(\tau_c - \varepsilon)} C_{l\phi}[\phi(\xi);t, \tau]\left(\frac{d\phi}{d\tau}\right)d\tau + \int_{(\tau_c - \varepsilon)}^{t} C_{l\phi}[\phi(\xi);t, \tau]\left(\frac{d\phi}{d\tau}\right)d\tau + \Delta C_{l\phi}[\phi(\xi);t, \tau_c] \tag{2.2}$$

where:

$$\Delta C_l = C_l[\phi(\xi);t,\tau_c + \varepsilon] - C_l[\phi(\xi);t,\tau_c - \varepsilon] \tag{2.3}$$

If there are no critical state encounters in the time interval 0 to t, integration over the complete interval is permitted and ΔC_l is identically zero. Note that this is entirely consistent with setting $d\phi/d\tau = 0$ for $\tau > \tau_c - \varepsilon$ in the equation above. The remaining non-zero terms are the first two. Furthermore, the upper limit of the surviving integral term can be replaced by "t" since the integrand is identically equal to zero for $\tau > \tau_c - \varepsilon$. Myatt and Jenkins have recently identified the transition model and described the regression-based methods to define the model's parameters (Ref. [1.16])

Tobak and Schiff, (Ref. [1.21]),suggested possible simplifications to the functional representations for indicial responses on the assumption that the motion, $\phi(\xi)$ is mathematically analytic over the interval $-\infty < t < \tau$, where τ is defined in Equation 1. This allows $\phi(\xi)$ to be replaced by its Taylor series expansion about $\xi = \tau$ because the derivatives of all orders for $\phi(\xi)$ must exist. Thus, $\partial C_l/\partial\phi$ can be written as:

$$C_{l\phi}[\phi(\xi);t,\tau] = C_{l\phi}(\phi(\tau), \dot{\phi}(\tau), \ddot{\phi}(\tau), \ldots;t,\tau) \qquad (2.4)$$

On physical grounds, the "distant" past can be expected to be less important to the step response than the motion characteristics just prior to onset, leading to the conclusion that only a few Taylor series coefficients need to be retained. Furthermore, for sufficiently slow reference motions, the generalized superposition integral can be approximated by an asymptotic expansion. Jenkins has shown that useful relationships exist between steady-state oscillatory force and moment data and the indicial response parameters as result of these observations, simplifying the task of formulating and identifying the NIR parameters (Ref. [1.17]). It should be noted that these parameters can be obtained from numerical computations, experimental tests, or by analytical means, whichever is appropriate or available. Our regressive analysis methods will be similar to those described by Reisenthel (Ref. [1.18]) with the additional requirement that terms for blowing effects be included. This proposed nonlinear indicial prediction model requires large amounts of unsteady aerodynamic data, both with and without blowing. The method for gathering this data is discussed in the following section.

It is proposed that the concept should initially provide accurate predictions of the aerodynamic loads and responses for a single degree-of-freedom, namely roll. The effect of blowing will then be added to the model before it is further extended to a second degree-of-freedom in yaw, with and without blowing. This approach offers the lowest risk for a high return in fundamental insight into the physics of the flow. A review of the theoretical and experimental work to date would indicate that NIR-based prediction models have been attempted only for single degree-of-freedom systems, with the degree-of-freedom being either roll or pitch. These models also do not incorporate the presence of tangential blowing, which introduces additional bifurcations and jumps to the solution (Ref. [1.13] toRef. [1.17] and Ref. [1.21] to Ref. [1.22]). Furthermore, no attempts have been made to develop control laws using NIR-based models (Ref. [1.22]).

The impact of blowing on the vortical flowfield is understood to be nonlinear with respect to the roll and yaw moments and is characterized by discontinuities. The control strategy formulated by Pedreiro for roll-yaw regulation consisted of blowing a minimum amount of air $C_{\mu_{min}}$, plus an amount, ΔC_{μ}, as determined by the control logic. This approach avoids the region where the values of C_{μ} are small. This region is where blowing is a highly non-linear actuator. In the jargon of NIR-based models, the region for small C_{μ} $(-0.02 \le C_{\mu} \le 0.02)$ has a minimum of 5 critical states, by inspection. This implies that the addition of a small amount of blowing to a vortical flowfield affects it substantially, leading to the loss of equilibrium-flow stability at specific values of blowing jet momentum. There are subregions within the region

$(-0.02 \leq C_\mu \leq 0.02)$ where it would appear that incremental changes in the amount of blowing have a strong and linear effect on the roll and yaw moments, as expressed by the large and analytic gradients in Figure 1.2. This observation would lead to the conclusion that the equilibrium flow in these subregions is asymptotically stable to small perturbations in the amount of blowing. This implies that an NIR-based model might prove useful as a tool to describe the impact of blowing on the yaw and roll moments for the region $(-0.02 \leq C_\mu \leq 0.02)$. This would allow for the generation of control laws that use a minimum amount of blowing. However, it should be noted that this region is very small, requiring precision in both measurements and blowing valve operation.

The sensitivity of the NIR-based model's accuracy to precise experimental measurements is further complicated by the current geometry of the fore-body of the wind tunnel model, which is a circular cone. Observations made both at Stanford and elsewhere (Ref. [1.23] and Ref. [1.24]) have shown that this particular geometry is susceptible to flow asymmetries which are sensitive to the micro-asymmetries of the nose. Should small changes in the nose geometry occur, the required level of measurement precision would be compromised. Pedreiro overcame this problem by consistently "fixing" the nose throughout all the experiments. Research done on the subject suggests that changing the nose geometry to one with a chined cross-section or a circular cone with a "blunt," rounded tip would alleviate asymmetric flow and provide a more stable flow condition overall by reducing the flow's sensitivity to changes in the nose geometry (Ref. [1.9] and Ref. [1.10]). Investigations of possible modifications to the nose geometry will be carried in the wind tunnel with nose sections of various geometries. Current candidates include a "shark-nosed-tip," a circular nose with strakes and a blunt circular cone. This work will be completed prior to the start of the experimental work detailed above.

2.1.4 Research Activities Proposed for 1996-1997

The following tasks are proposed to develop our understanding of the mechanisms through which tangential fore-body blowing works and explore the feasibility of its use to control and/or improve the motion of an aircraft at high angles of attack:

(1) Complete the investigation into possible modifications of the nose geometry to improve flow symmetry, which is strongly affected by micro-asymmetries at the nose tip. Implementation of a modified nose tip will also increase the repeatability of the vortex flow structure by fixing the line of separation at and around the nose.

(2) Develop an aerodynamic model to provide a more complete description of the coupling at high angle of attack between aircraft dynamic motion, the aerodynamic forces and moments generated by the vortical flowfield and the effects of FTB.

2.1.5 Research activities planned beyond 1996-1997

(1) Formulate control laws based on the aerodynamic model to allow for the capability to command large roll and yaw angles at high angle of attack using FTB as the actuator.

(2) Experimentally demonstrate the capability to command large roll and yaw angles at high angle of attack using FTB in the wind tunnel.

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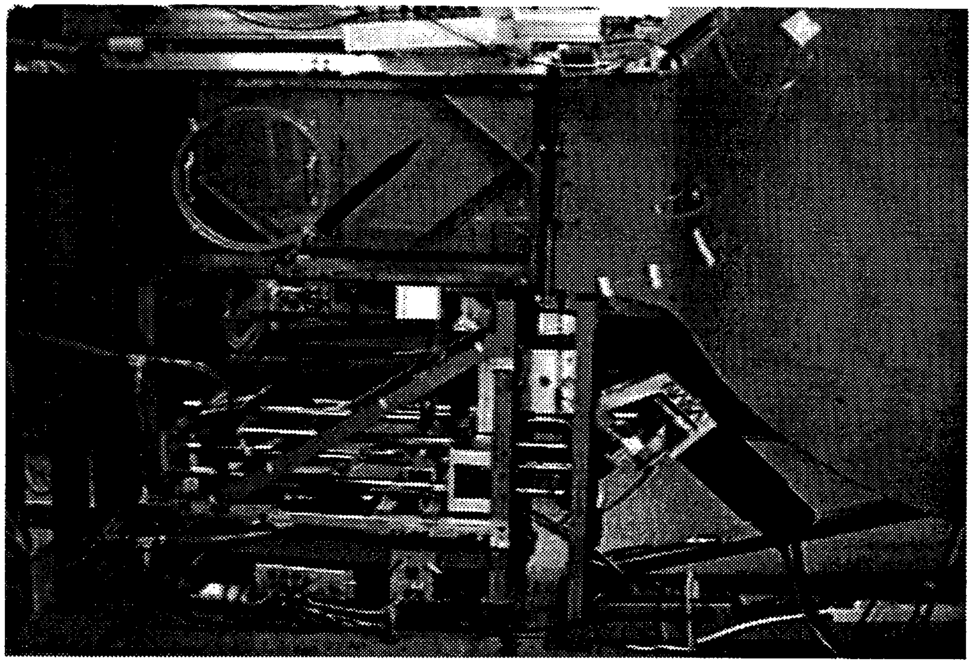

Figure 1.1 Wing-rock Model in the Stanford Subsonic Windtunnel

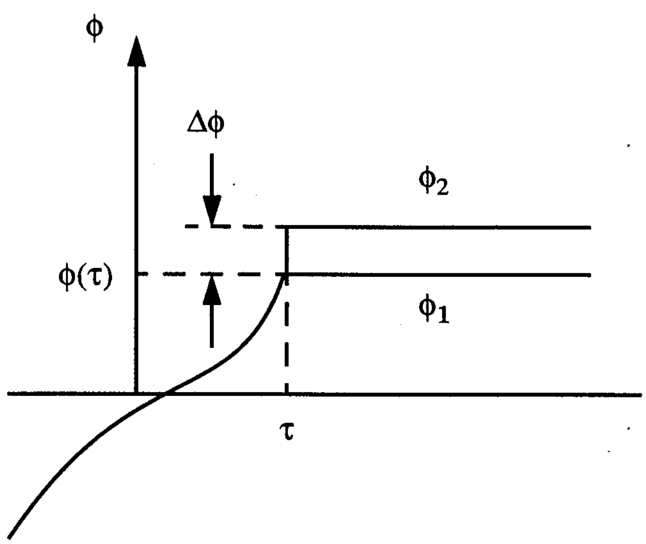

Figure 1.2 Nonlinear Indicial Response Motions

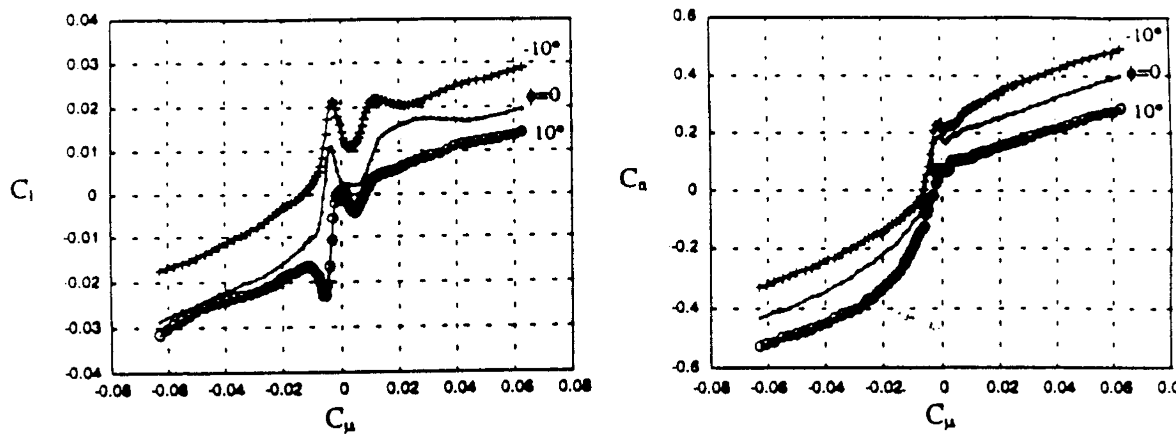

Figure 1.3 Static Roll and Yaw Moment Coefficients (C_l and C_n) versus the Jet Momentum Coefficient (C_μ) for various ϕ and $\gamma = 0$. $\alpha = 45$ degrees, U_∞=19.5 m/s

2.2 PROJECT 2 - LES AS A TOOL FOR STUDYING JET AERO-ACOUSTICS

Research Participants: Prof. S. Lele, graduate student

Ames Technical Contact: Dr. N. Mansour

2.2.1 Introduction

The prediction and control of aerodynamically generated noise is important to the design of quieter subsonic and supersonic aircraft. Current prediction methods primarily rely upon empirical formulae and the use some form of acoustic analogy (pioneered by Lighthill) for scaling data. The flexibility of such tools is limited by their empirical origins. With the recent advances in computational algorithms and computer hardware, a new generation of analysis tools can be developed which have a theoretical basis. For aero-acoustics, Large Eddy Simulation (LES) holds the promise of predicting the dominant features of noise radiated to the far-field by a flow such as a jet issuing from a nozzle.

Predicting the far-field noise via LES is far more challenging than an overall prediction of the near-field aerodynamic flow. Acoustic predictions are dependent upon a two-point space time correlation of flow quantities (Ref. [2.2]), a far more demanding test of the flow prediction's fidelity. By its nature, LES does not resolve all the dynamical scales of the flow being computed. The accuracy of LES results depends on the sub-grid scale models employed, which represent the range of scales not resolved in the simulation, and on the numerical algorithms being used (discretization scheme, time advancement, numerical boundary conditions, etc.). Recent developments in sub-grid scale models (Ref. [2.3]), while very attractive and promising for aerodynamic predictions, have not been tested from the viewpoint of far-field noise prediction. The present research is directed at extending the basic LES formulation to the prediction of far-field noise and testing how well such a method performs.

2.2.2 Research Objective

Under the premise that the dominant features of the far-field noise are associated with the dynamics of the energy containing range of scales in the near field aerodynamic flow, a Large Eddy Simulation which captures these energetic scales can be expected to contain sufficient near field information to predict the far-field noise. The proposed research focuses on determining LES's efficacy in making acoustic predictions, considering the effects of sub-grid scale models.

Current sub-grid scale models which attempt to parameterize the influence of the unresolved scales have been developed to allow an overall statistical prediction of the aerodynamic near-field. The models are typically designed to provide correct energy transfer between the resolved and unresolved scales. While these models seem quite promising, their impact on far-field acoustics has not been considered. Since the far-field noise is a very small by-product of the flow, it is necessary to ensure that the sub-grid models do not behave as a low order (and hence efficient), but spurious source of sound. As Crighton (Ref. [2.1]) points out, sources of this type may be introduced via discretization errors

and numerical boundary conditions, etc. It is necessary to examine sub-grid scale models in this context. Furthermore, for practical LES applications, the sub-grid scale energy may be as much as 10-30% of the resolved energy and this may require that closer attention be paid to the acoustic sources implied by such models.

2.2.3 Research program

We propose to carry out a program of research aimed at applying the LES methodology from the point of view of far-field noise prediction. The fidelity of LES in predicting the unsteady flow and acoustic sources will be judged by making extensive comparisons with a Direct Numerical Simulation (DNS) of the same flow configuration. For this reason, the first phase is a direct simulation of turbulent flow in simple geometries.

A formulation capable of yielding a stationary turbulent jet flow while maintaining the efficiency and accuracy of spectral methods was developed. The method is an extension of Spalart's method (Ref. [2.5]) for simulating boundary layers, to the case of a co-flowing jet or wake. The results resemble those obtained by Spalart (Ref. [2.6]) for the sink-flow boundary layer. The method involves incorporating the slow spatial growth effects via a decomposition of the variables according to their multiple spatial scales and a suitable coordinate transformation (Ref. [2.7]). The derivation is more rigorous than the boundary layer analysis, due to the simplification introduced by explicitly considering the small deficit limit.

The result of the formulation is a modified set of equations consisting of the Navier Stokes equations with an additional set of small growth terms. The implied flow field is homogeneous in both the stream-wise and span-wise directions, and was therefore implemented in the spectral code used by Rogers and Moser (Ref. [2.4]). Testing of the code's ability to maintain a stationary flow was carried out, and a preliminary Direct Numerical Simulation was completed.

2.2.4 Research activities proposed for 1996-1997

The DNS will first be extended to collect the data necessary to make acoustic predictions. This will involve continuing the computation from the final condition previously reached and computing the two point space-time correlations necessary for acoustic predictions. Concurrently, the code will be modified to perform the Large Eddy Simulation of the same flow field. The LES calculations will begin with the simplest Smagorinsky type sub-grid scale eddy viscosity models. Comparisons of the LES and DNS results will begin with one point statistics and move on to the two-point correlations mentioned previously. If the LES and DNS results compare well at this level it may be concluded that the dominant acoustic sources have been effectively modeled (at least in the context of an acoustic analogy).

2.2.5 Research Activities Planned Beyond 1996-1997

Once the initial tests of LES's fidelity have been carried out using a simple sub-grid scale model, the direct impact of different sub-grid scale models will be examined. This will require a study of the space time correlation of the model sub-grid stresses and the resolved stresses in the DNS database. It is expected that this will involve reintegrating the DNS data from the coarsely spaced times available as restart files. When the sub-grid scale energy is non-negligible these correlations may provide information about how much noise is radiated by the unresolved sub-grid scale motions and how effectively it is captured by sub-grid models employed in the calculations. Once the appropriateness of LES has been established, the prediction of far-field noise which uses the near-field unsteady flow calculated via LES can be applied to flows of engineering interest. LES calculations of an experimental flow for which detailed flow and noise data exist may be the next logical step in this direction.

2.2.6 References

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2.3 Project 3 - Research on a Lifting Wing-Flap Configuration

Research Participants: Prof. B. Cantwell, graduate student

Ames Technical Contact: Dr. L. Olsen, Dr. J. Ross

2.3.1 Introduction

Experiments by Kendall and his co-workers (Ref. [3.1], Ref. [3.3] and Ref. [3.4]) and Grosche et al. (Ref. [3.2]), using a directional microphone to measure noise distributions on a wind tunnel model, found that the noise generation was highly localized. The most active locations were the wing tip and leading edge, the trailing corner and trailing edge of the flap and the gap separating adjacent flap elements. Areas of attached turbulent flow were inconsequential noise sources. In experiments with individual flaps, vortex roll-up was postulated as the major reason for noise generation. When the vortex strength was reduced, the noise intensity decreased. Kendall observed that the major part of the noise generation was caused by the gap between two differentially deflected flaps. Kendall also argued that the trailing edge noise did not play an important role.

Recent experiments (Ref. [3.5] and Ref. [3.6]) on a wing-half span flap configuration by NASA investigators in the Ames 7 x 10 included a collaborative program of noise measurements by Boeing researchers. As with Kendall's findings, the flap edge was clearly identified as the major source of noise. Various edge treatments were tested in an attempt to reduce noise. Large differences in the effectiveness of noise reduction were observed depending on the particular choice of flap edge treatment.

A Fluid dynamic description of noise generation from a lifting surface is extremely complex: confluent turbulent boundary layers and vortex sheets roll over edges producing large surface pressure fluctuations. The flow involves a wide range of length scales, high local shearing stress and intense turbulence activity over the lifting surface. A better understanding of these processes is needed for the development of effective methods for airframe noise reduction.

2.3.2 Research objective

The objective of this research is to understand the flow mechanisms responsible for noise generation by a wing and trailing edge flap combination. A NACA 63-215 Mod B airfoil section with a Fowler Flap has been selected for flow measurements in the areas of high noise generation. For comparison purposes, the same model geometry used in the Ames 7 x 10 experiments will be used in computations by Stanford and Ames investigators and in the small scale experiments at Stanford. A separate set of hot wire measurements will be undertaken in the 7x10 to determine local turbulence intensities and scales in the near wake at the higher Reynolds number available with this facility. The facilities at Stanford University and NASA-Ames allow the flap edge flow field to be studied over a range of Reynolds numbers from 50,000 to 2.5×10^6.

2.3.3 Research program

The identical model geometry was selected for experiments both at Stanford and NASA-Ames and also for the computational work. The model is designed with two interchangeable middle parts to allow for instrumentation suited to either the wind tunnel or the water channel environments. For the wind tunnel experiments, the middle section is instrumented with pressure tappings and embedded pressure sensors. In this configuration, the model as a whole contains 35 pressure tappings on the main section and three pressure sensors on the side of the semi-span flap. For the water channel experiments, the middle section is replaced with a geometrically identical section containing dye ports instead of pressure tappings. The main wing has 12 dye ports; four on the top surface, seven on the bottom and one at the edge of the cove section. Of three dye holes on the flap, one is located on the top surface while the other two will be used to inject dye on the lower surface.

Early this year, the unassembled semi-span flap model was delivered from the machine shop. This model was checked for fit and found satisfactory. For the initial wind-tunnel flow visualization experiment, the model was assembled in it's dye injection configuration. It was felt that the section incorporating dye ports would better withstand the rigors of surface oil-flow measurements. Upon installing the model in the wind-tunnel, it was found that the original mounting arrangement was unsatisfactory from both a stiffness and a positioning perspective. A new mount was fabricated with the ability to repeatedly set the flap gap and overlap to within a few thousandths of an inch. Figure 3.1 shows the model installed in the test-section ready for an experiment.

Preliminary measurements were made using an uncalibrated hotwire and an IBM- PC based data-acquisition system. Shortcomings in the usability of this system made it desirable to complete rewrite the data-acquisition software to one based around the LabView programming language. Changing to the LabView system allowed one computer to simultaneously acquire data and control a three-axis precision positioning system. This positioning system, in conjunction with the hotwire (or 5-hole probe) allows for extremely fine resolution traverses to be made of the wake plane. A plot of three of these scans made with an uncalibrated hotwire is given in Figure 3.2. On the left is the mean hotwire signal, and on the right is the RMS value. The plots are shaded by magnitude, and are ordered from top to bottom by distance from the trailing edge of the flap - the first at 0.5 inches, the two subsequent at 3 inch intervals. For these experiments, the main element was at a zero degree angle of attack; the flap deflected down at 20 degrees.

A second view of this data was also found to be instructive. From the three-dimensional perspective of Figure 3.3 in a plane 0.5 inches from the trailing edge, it is quite easy to see the vortex roll-up over the edge of the flap, the velocity jet in the center of the vortex, and the wake of the main element merging with the wake of the flap as it is drawn over the deflected flap surface. The same

view in Figure 3.4 illustrates that further downstream of the flap edge (approximately 6 inches), the jet at the vortex center becomes a velocity deficit - suggesting a strong negative pressure gradient in this direction.

As these experiments were conducted at a Reynolds number of 600,000 with no roughness elements affixed to the model, there was some concern that the flow might be separated. To resolve these questions, two experiments were conducted - one using a surface oil-flow technique, and one using fine tufts of thread attached to the model. Both of these investigations supported the conclusion that at a zero-degree angle of attack on the main element, the flow remained attached over the flap until the flap deflection exceeded 40 degrees. It was also established that attachment was relatively insensitive to variations in flap gap and overlap.

2.3.4 Research activity proposed for 1996-1997

To begin the new academic year, several improvements to the tunnel are contemplated. It has been discovered that at the full tunnel speed, there is a significant temperature rise over the course of an experiment. To this end, a heat exchanger will be added. Further, as some of the experiments run for more than one day, dynamic pressure feedback will be incorporated into a new motor speed controller so that a constant 'Q' can be maintained. We would also like to increase the horsepower of the motor currently in the tunnel to achieve velocities on the order of 33% higher. It is hoped that all of these modifications can be accomplished in the first part of the year.

Experimentally, a clean tunnel survey needs to be done to establish the flow angularity and uniformity in the inlet and exit to the test-section. As a baseline, this data will be taken with a 5 hole probe. A hotwire survey of the tunnel needs to be preformed to measure the turbulence levels. If the turbulence is unacceptable, several options have been considered to smooth out the flow upstream - either by adding screens, flow straighteners or pressure plates. The hotwires in use now are uncalibrated. While this is sufficient for qualitative data, it is not appropriate if the results are to be compared with a CFD dataset. Much effort needs to be made towards developing a device for automated hotwire calibration. Several methods are under consideration - including the 'dynamic calibrator' idea used A.E. Perry and his group at the University of Melbourne, or a more traditional static calibration by varying the windtunnel speed and altering the angle of attack of the probe. Measurements will be made using a previously calibrated five hole probe. This same five-hole probe will be used to duplicate the three wake cuts shown in Figure 3.2 at a variety of tunnel speeds and flap rigging settings. The flap rigging settings will be as close a possible to those used in the Ames 7x10 experiments made with the identical larger scale model. Once the 5-hole probe surveys have been accomplished and the hotwire calibrated, the double hotwire will be used in the flap edge region to map out the Reynolds stress. These measurements will be supplemented with data from pressure sensors buried within the wing to record the fluctuating surface pressure.

2.3.5 Research activity planned beyond 1996-1997

Eventually we would like to make use of the Luminescent Paint technique discussed in section 2.4. We hope to make pressure sensitive paint measurements on the wing to provide a more complete picture of the mean pressure field on the wing at various flap settings. Fluorescent dye visualization of the flap edge flow will be carried out in a water channel by replacing the pressure sensor portion of the model with a section that has been fabricated with dye ports.

A concurrent effort is underway to compute the same flow measured experimentally in these tests. It is the ultimate intent of this research to compare the results from these two data-sets and from recent acoustical surveys done at Ames to look for correlations that would suggest a tie between the noise generated, and the underlying physics of the local flowfield. This knowledge would have application as a design tool to quantify the effect that the flap edge vortex has on noise generation.

2.3.6 References

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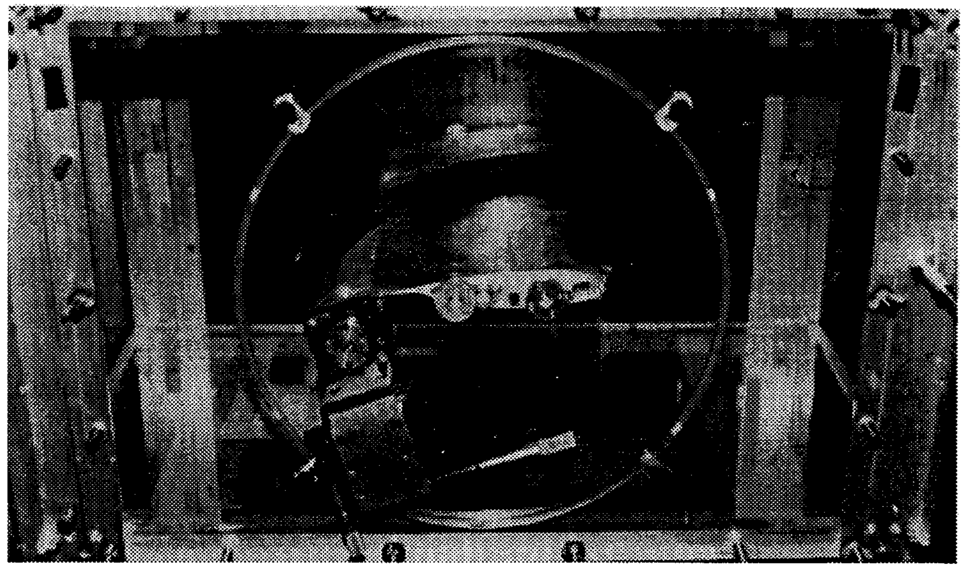

Figure 3.1 Semi-span Flap Model in Test-section

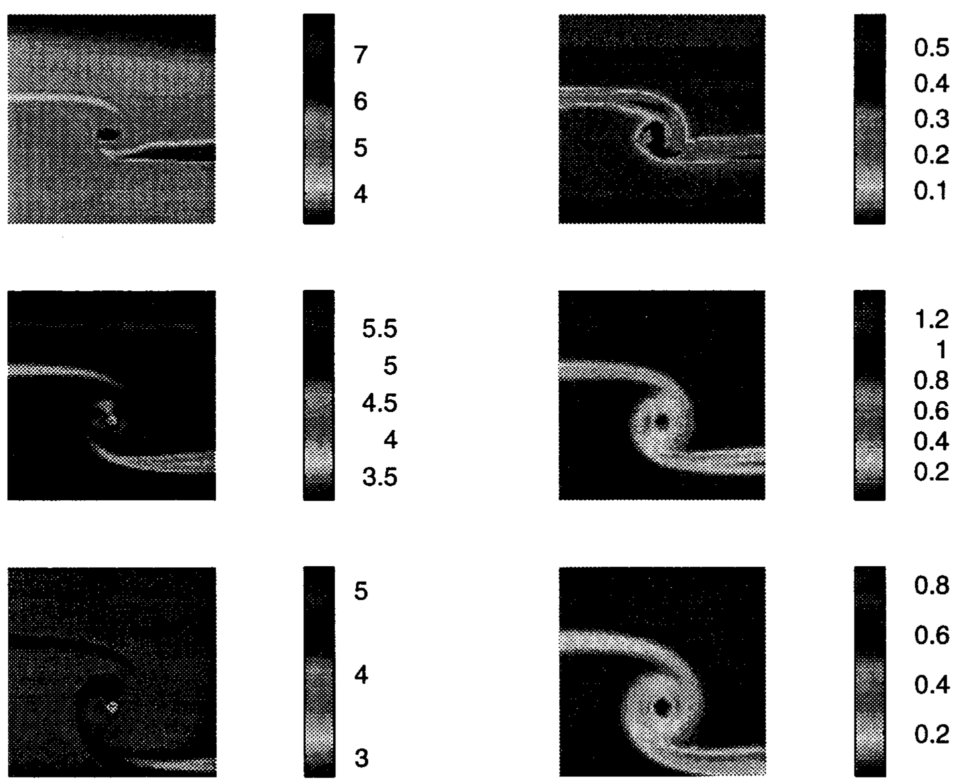

Figure 3.2 Wake Survey of Deflected Flap

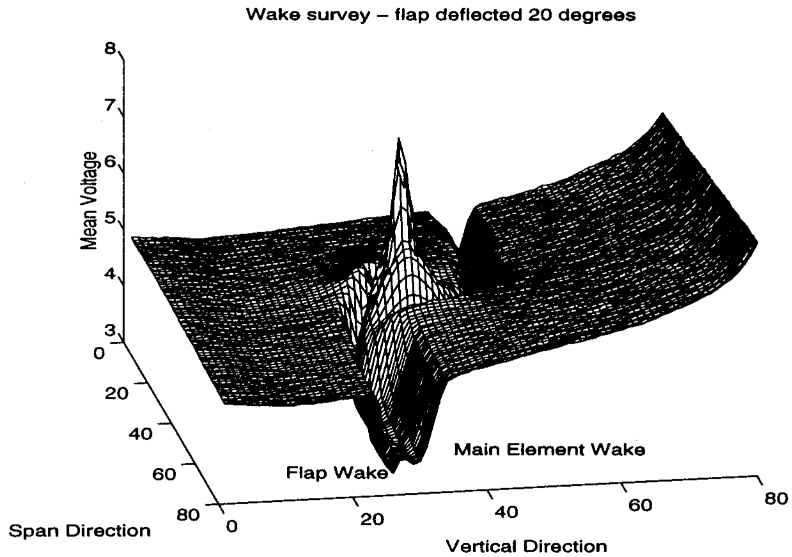

Figure 3.3 3-d Wake View, 0.5 Inches from Trailing Edge

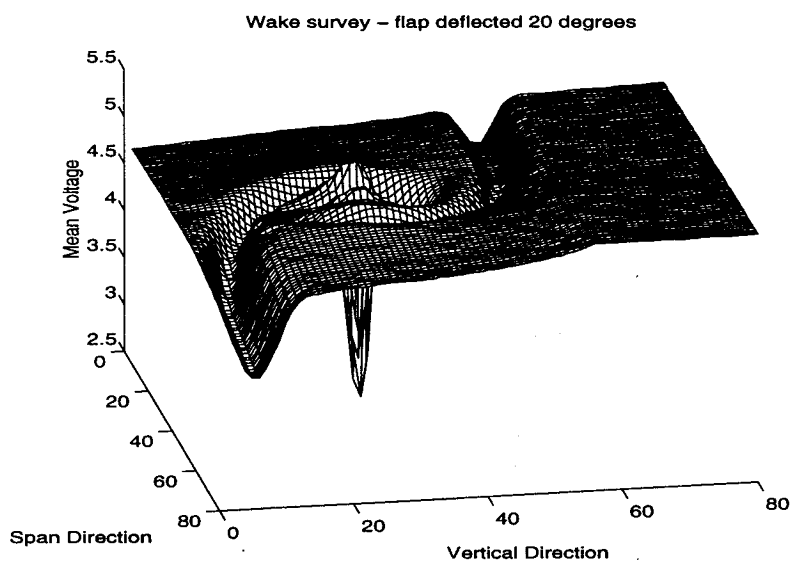

Figure 3.4 3-d Wake View, 6.5 Inches from Trailing Edge

2.4 PROJECT 4 - LUMINESCENT PAINT FOR AERODYNAMIC MEASUREMENT

Research Participants: Dr. R. Mehta, Graduate Student

Ames Technical Contact: Dr. S. Davis

2.4.1 Introduction

Pressure sensitive paints (PSP's) are now used routinely for measuring surface pressures on wind tunnel models at transonic and supersonic Mach numbers. This measurement technique utilizes a surface coating containing fluorescent or phosphorescent materials, the brightness of which varies with the local oxygen partial pressure on the surface. In current practice, a wind tunnel model is coated with the PSP, which is then illuminated with light of an appropriate wavelength to excite the material. The illuminated model is imaged with a digital CCD camera during the wind tunnel test. The images are then computer-processed in order to obtain a map of the surface pressure distribution. The relationship between surface brightness and pressure is generally determined by calibrating the paint (in situ) using a few pressure taps on the model. PSP's have some important advantages over pressure taps which are typically used for surface pressure measurements on wind tunnel models, as the paint provides a measurement over the entire model surface. Measurements using PSP's hold the potential of being much less expensive (over time) than those with pressure taps.

2.4.2 Research Objective

Low-speed ($M < 0.1$-0.2) aerodynamic testing is becoming increasingly relevant. For example, complex multi-element systems are being designed for subsonic and supersonic transports for the take-off and landing phases which must be extensively studied and tested in wind tunnel experiments. Experiments using the PSP technique have the potential of providing the surface pressure profile over the entire model in a relatively fast and inexpensive manner. However, the one limitation of PSP's is that the brightness of the paint is inversely proportional to the pressure. Therefore, detection of differences in brightness, which relate to differences in pressure on the model, become increasingly difficult as the flow speed is reduced. Although we had some recent success with low-speed (~ 35 m/s) testing of a delta wing model (Ref. [4.1]), it is clear that further work is needed in order to improve the accuracy of the technique and to make it more versatile. This involves identifying paints which have a higher sensitivity to pressure, and in addition, a lower sensitivity to temperature.The overall program objective is to apply the luminescent paint technology to the study of basic fluid physics problems, especially at subsonic speeds.

Recently there has been some progress in the development of paints with improved time response. We intend to continue Shimbo's work (Ref. [4.2]) on the application of PSP's to unsteady flows. A problem of particular interest

would be the application of PSP's to the study of the wing rock phenomena. In this problem the frequencies of interest are low enough so that current paint responses should be adequate to resolve the time-dependent surface pressure.

2.4.3 Research program

The portable image acquisition and reduction system for the Pressure/Temperature Sensitive Paint program was upgraded. A Pentium based 133 MHz computer, with 32 MB of RAM was installed. In addition to the ability to collect and manipulate large amounts of data, the operations now run in a fraction of the time it took previously. During the first part of last academic year, time was devoted to bench testing various PSP samples obtained from Purdue University. The bench tests consisted of an inclined turbulent jet impinging on a flat plate with the paint sample applied to it. The main goal of the bench tests was to identify paints which would be more suitable for low-speed ($U_e \sim 30$ m/s) aerodynamic testing. The initial results indicated that some of the ruthenium based paints responded extremely well to very low pressures. However, further, more quantitative, investigations showed that the paint was also responding to some other environmental influence, most likely moisture in the shop air supply. At that point, these new paint samples were abandoned in favor of the original (well tested) molecule, PtOEP.

The later part of last academic year was spent on testing temperature sensitive paints for transition detection on a 10 degree cone model installed in the Fluid Mechanics Lab Supersonic Quiet Wind Tunnel. The purpose of the test was to identify transition (actually the lack of) on the cone model to support the hypothesis that the flow was indeed quiet in the test section at the operating Mach number of 1.6. The initial tests were conducted with a readily available (Rhodamine B) temperature sensitive paint. While the paint responded adequately to temperature changes in the bench tests conducted at room temperature, the same calibration did not hold at the wind tunnel operating temperature of about $-20\,^{\circ}$C, and it was therefore deemed unsuitable for the purposes of detecting transition. In the current series of tests, a paint based on the europium molecule (EuTTA) is being used. Calibrations performed at Purdue University show that this paint responds linearly, with an acceptable sensitivity of 1.5% change in intensity ratio per degree C, over the temperature range of interest ($T = \pm 20\,^{\circ}$C). Some calibrations in the Supersonic Wind Tunnel confirm this sensitivity. This paint was first used to detect transition on a F16XL wing model. Although some promising results were obtained, the flow on this model appeared to be rather complex and so some further testing on the cone model is now proposed.

We are currently involved in obtaining some PSP measurements on a high aspect ratio wing in the High Reynolds Channel. This test is part of the Advanced Subsonic Initiative on Wing Design and the main objective it to generate data for turbulence model validation. Since the High Reynolds Channel is pressurized with the static pressures in the shell approaching 100 psi, a spe-

cial pressure vessel was designed and constructed to hold the digital camera. A new PSP (PtTFPP/FIB7), developed at the University of Washington, is being used for this test. This paint has a slightly lower temperature sensitivity and it also photodegrades at a lower rate. The preliminary testing phase has just been completed and the initial data sets look very promising. This is the first time that the pressure paint technique has been applied at such high pressures.

The former graduate student on this project, Yuichi Shimbo, spent the year developing the PSP capability at Stanford University, using a standard (8 bit) video camera. A parametric study of an impinging jet was conducted and the time response of the paint (PtOEP) was also evaluated. In the final phase, data were obtained using the Ames 14 bit camera on a delta-wing model in the Aero/Astro low-speed wind tunnel. At angles of attack of more than 20 degrees, the secondary separation line was clearly visible in the paint data at a free-stream velocity of 35 m/s ($C_p \sim 1.5$-1.8) and at 30 degrees angle of attack, vortex breakdown was also evident (Figure 4.1). The effects of span-wise blowing were also investigated. Since better resolution is required for these low-speed studies, the FML digital camera and computer system were used for these experiments. These results will also be presented at the AIAA Meeting in Reno (Ref. [4.1]).

2.4.4 Research activities proposed for 1996-1997

Testing of the High Aspect Ratio Wing in the High Reynolds Number Channel is due to start very shortly. PSP data will be obtained on the model using the PtTFPP/FIB7 paint combination. The data will be converted to pressures using in situ calibrations based on pressure tap readings. These pressure measurements will form part of an overall data set consisting of skin friction measurements and velocity profiles obtained using a 3-D LDV system.

Temperature sensitive paints will be used to conduct further transition studies in the Quiet Supersonic Wind Tunnel, both on the cone model and the F16XL wing model. The initial task is to ensure that the paint is responding in the predicted manner at very low temperatures ($T \sim -20\,°C$).

Efforts will continue to improve the low-speed testing ability with PSP's so that the relatively small pressure differences may be better resolved. Some newer paints which are less temperature sensitive should help in this effort. Relatively complex aerodynamic configurations such as wing/body junctions and multi-airfoil systems will then be studied using this technique. We plan to build on the work begun by Yuichi Shimbo who studied the response of PtOEP to a periodically changing pressure with a period of 12.8 s. During the 1996-97 year we will attempt to extend Shimbo's first order model of the time-dependent paint response and to develop a better fundamental understanding of the physics of the paint-substrate response. In addition we will continue to seek faster responding paint materials.

In a bench-top experiment, the flow field on a flat plate with an inclined jet impinging on it will be studied. The plan is to obtain a complete map of the surface shear stress field (magnitude and direction) using the liquid crystal technique being developed by Dan Reda in the FML, and the corresponding pressure field using PSP's.

2.4.5 *Research activities planned for beyond 1996-1997*

Eventually one of the most beneficial applications of PSP's will be in the area of unsteady flows. If the efforts to identify faster responding paints are successful, we intend to use PSP's to study the unsteady surface pressures on a wing model subject to wing rock instability.

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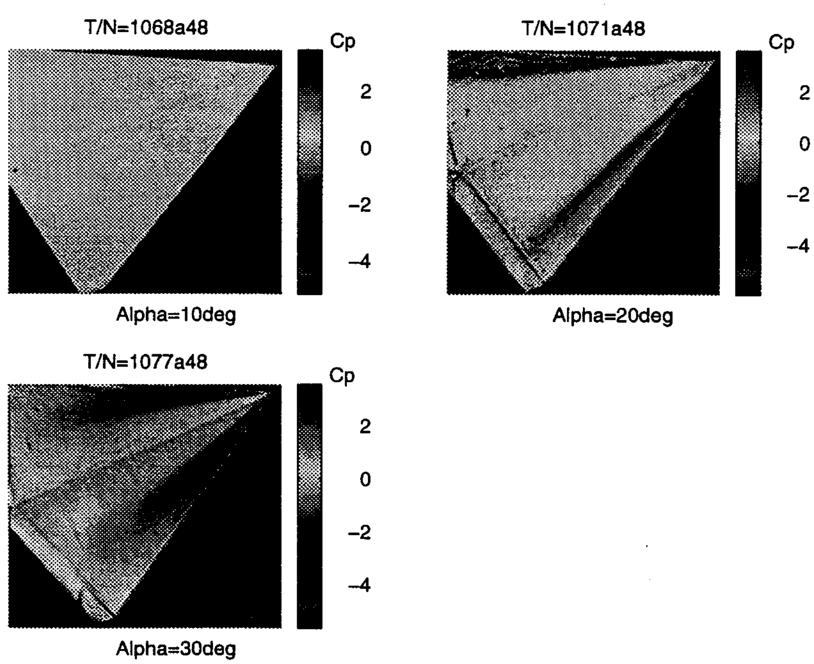

Figure 4.1 PSP response for increasing angle of attack (β = 0 deg, U = 35m/s, C_μ =0)

2.5 Project 5 - Prediction of Wing Maximum Lift for Preliminary Design

Research Participants: Prof. I. Kroo, Graduate Student

Ames Technical Contact: Dr. L. Olsen, Dr. S. Smith

2.5.1 Introduction

Traditional aircraft selection methods, such as parametric studies and summary charts, allow the designer to pick the "best" design based on variation of a limited number of parameters. This best design is, however, still sensitive to many other variables that were not examined in the original parametric studies. The true optimum design can often only be found when all of the available design parameters are varied simultaneously. Recent advances in the field of multidisciplinary optimization make this approach promising for use in the conceptual design of new aircraft. The major benefit of computerized optimization for aircraft design is the ability to perform these trade studies with many more parameters and much greater speed than can be accomplished through traditional methods.

While numerical optimization provides the benefit of a more thorough exploration of the design space, it also presents new challenges for the modeling of the aircraft throughout that design space. In order to best meet their objectives, optimizers tend to push designs to their limits. This sometimes results in exploitation of weaknesses in the aircraft model, producing designs that are not practically feasible. For this reason, one of the most important issues in using optimization for aircraft design is accurately modeling the various effects that drive the design. These models must also be simple enough to run quickly since an optimizer may require thousands of function evaluations to explore the design space. There is a fundamental trade-off between accuracy and speed that must be properly made in order to formulate analysis techniques that make numerical optimization a practical tool for use in conceptual and preliminary design.Wing maximum lift is one of the areas that has been most difficult to model accurately for optimization. It has also been shown to be a very important parameter for choosing optimum wing planforms (Ref. [5.1]), due to significant effects on aircraft noise, cost and performance. The trade-off between high sweep for low drag at high Mach numbers and low sweep for good low speed performance and handling qualities is of fundamental importance. Unconventional aircraft designs, like the blended wing-body configuration, may also have wings with sharp changes in sweep and thickness at various span-wise stations that could have significant effects on the high lift aerodynamics of the wing. However, current methods used to evaluate wing maximum lift in conceptual design phases may not be sufficient to accurately model the effects of sweep and geometric irregularities.

The ideal model to determine wing maximum lift would be a 3-d Navier-Stokes code with a grid density fine enough to capture the details of the stalling phenomenon on the wing. The computational time required for such a Navier-Stokes calculation, is prohibitively expensive for use in an optimization algo-

rithm (in fact, it is not compatible with any of the computers available at the current time). Recalculating an entire 3-d Navier-Stokes solution for each combination of the many design parameters chosen by the optimizer would be impractical.

The starting point for the research described here is a wing model developed by Sean Wakayama at Stanford University (Ref. [5.1]). This method uses a critical section analysis which compares local section lift coefficients, calculated from a Weissinger vortex lattice method, with estimates of 2-d maximum section lift coefficients based on empirical data. Flaps are simulated by increasing wing incidences in the Weissinger model, applying an increment in $c_{l,max}$ due to flap deflection on the flapped portion of the wing, and increasing the assumed $c_{l,max}$ due to induced camber on the sections near the flap edge. Finally, the maximum 2-d section lift coefficients are reduced by a factor of cos Λ as an empirical correction for the effect of sweep on the pressure distribution. The wing is then assumed to be at its maximum usable lift when any section c_l reaches some fraction of its local $c_{l,max}$. The assumption of a cos Λ variation in $c_{l,max}$, as opposed to the simple sweep theory (Ref. [5.2]) assumption that $c_{l,max}$ decreases with cos 2Λ, is based on experimental observations (Ref. [5.3]). It has been observed that proper placement of fences and vortex generators on swept wings can yield $c_{l,max}$ values that closely approach the 2-d unswept values (Ref. [5.4], Ref. [5.5]).

The critical section method may be justified for unswept wings, but its validity is suspect for wings with significant sweep for several reasons. The method assumes acquisition of 2-d $c_{l,max}$ by placement of boundary layer control devices without actually specifying the location or geometry of such devices. Also, wing sweep changes the shape of the pressure distribution at fixed total lift, increasing the magnitude of the leading edge pressure peak. More importantly, the existence of transverse pressure gradients along a swept wing induces boundary layer flow in the span-wise direction. This span-wise flow increases the length over which the boundary layer develops, resulting in a weaker boundary layer toward the wing tip. In certain codes used for high lift design at Boeing (Ref. [5.6]), 3-d panel codes are coupled with 2-d boundary layer codes, thus capturing the correct 3-d pressure distribution, but still neglecting 3-d boundary layer effects. The effects of sweep on the transverse boundary layer development and the pressure peak are the primary areas of interest in the proposed research.

2.5.2 Research Objectives

Some optimization results using the current high lift analysis have indicated the possibility of problems due to inadequately modeling the impact of the various effects of sweep and geometric discontinuities on $c_{l,max}$. The optimizer tends to give wings a few more degrees of overall sweep than existing designs for the same missions. It has also favored highly swept wing tips. Information on the variation of $c_{l,max}$ with wing sweep is vital to performing planform opti-

mization. Usable maximum lift is a major constraint that limits the sweep of a wing. With optimization results favoring larger sweeps than would be expected, a better assessment of the penalties sweep will impose on maximum lift capabilities is needed. Therefore, we propose to continue a program of research to 1) determine the effects of wing sweep and other geometric properties on maximum usable section lift and 2) develop an aerodynamic model and optimization routine suitable for use in the early stages of conceptual design of airplanes.

2.5.3 Research program

Computational models are being developed and used to examine various inviscid and viscous phenomena that effect the maximum usable lift of wing designs, especially with regard to changes in wing sweep. The results of this study will be used to formulate an improved algorithm for optimization of wing planforms in preliminary design methods.

During the first year of this research program, studies were initiated in two main areas. The first study focused on transverse boundary layer development, and the second on the effect of sweep on the inviscid pressure distribution for a wing of fixed total lift. The transverse boundary layer development is studied using a 3-d, incompressible Navier-Stokes code (INS-3D) to compute the flow properties along a segment of a swept wing. The wing segment runs all the way to the boundaries of the computational grid, eliminating the need for wing tips or roots. The idea behind modeling only a section of the wing is to use the available computational resources as efficiently as possible. By eliminating the wing tip and root, more grid points can be concentrated along the section of the wing where the boundary layer development is examined. Additionally, the grid is relatively easy to generate, allowing fast and efficient parametric studies varying sweep and airfoil sections. The key element of the model is then applying appropriate boundary conditions to simulate the actual flow over a section of a swept wing with or without a boundary layer control device.

Several cases have been run using the model described above with some promising results. The code was able to converge using the unconventional boundary conditions and evidence of a growing span-wise boundary layer can be seen in the flow solution. Substantial span-wise flow and some outboard flow reversal could also be seen using particle traces. Also, a code to compute boundary layer characteristics in the chord-wise and span-wise directions was developed and tested on some of the preliminary test cases.

In addition to the viscous phenomenon discussed above, an inviscid effect commonly observed to cause separation is sonic flow at the pressure peak (Ref. [5.3]). The lift coefficient at which this sonic flow condition is reached is affected by sweep for a couple of reasons. Since the normal component of Mach number is reduced by sweep, the increment in velocity required to bring the flow to sonic conditions is increased. However, from simple sweep theory, for a fixed free-stream lift coefficient the effective section lift coefficient on a swept section must increase by a factor of $1/\cos 2\Lambda$. This results in a higher magnitude pres-

sure peak with increasing sweep for a given total wing lift. Since these two effects of sweep are competing, it is of interest to develop a model examining the effect of sweep on maximum lift limited by sonic velocity at the pressure peak. The model used to calculate this maximum lift condition is briefly described below.

The pressure coefficient corresponding to sonic conditions is calculated as an isentropic function of the normal component of the free-stream Mach number. This pressure coefficient is then converted to an 'incompressible' velocity increment using the Prandtl-Glauert or Karman-Tsien compressibility correction. The incompressible velocity distribution over the normal airfoil section is computed at two different angles of attack. Due to the linearity of the solution, the velocity distribution is then known for any angle of attack. This yields a closed form solution (quadratic in alpha) for the lowest angle of attack at which any point on the airfoil reaches sonic conditions, thus defining the maximum allowable angle of attack for the airfoil section. Then the maximum section lift limited by sonic conditions is calculated using this angle of attack and converted back to a compressible lift coefficient, again using the compressibility correction. The maximum lift based on free-stream conditions is the computed maximum section lift times the square of the cosine of the sweep angle (which assumes that Jones' simple sweep theory applies for this inviscid model.) This model provides a very simple and fast calculation of a constraint that is widely recognized as one of the primary factors limiting maximum usable lift, and it performs very well in comparison with results from full potential code solutions over a wide range of angle of attack and Mach numbers for various airfoils. A more detailed discussion of both the transverse boundary layer model and the sonic pressure peak model can be found in the 1994-95 JIAA progress report.

With this preliminary work done on modeling some boundary layer flow and maximum lift criteria, the next step in this research program was to develop a simple 3-d aerodynamic code to use as a basis for the wing design/optimization program. The criteria for this model were that it be simple and fast while giving a fairly accurate 3-d pressure distribution over general wing geometries. Simplicity in both defining the geometry and solving for the 3-d flow were considered important features for the desired application. Since the primary focus of this research is low speed, high lift flight, a linear aerodynamic panel method was chosen. There are several reasons for developing an original code here rather than using some existing panel code. Most importantly, this code will eventually be coupled with a boundary layer code and an optimizer, all of which can work more efficiently if they are designed together, with the developer having a complete understanding of how each one works.

As a preliminary step in the design of the panel code, several 2-d experiments were performed to examine ways of simplifying the 3-d method. These 2-d experiments proved very helpful, and eventually led to the selection of the algorithm for the 3-d aerodynamic model. Some of these experiments are described below. The first method examined is a distributed source and vortex model with

singularities placed on the airfoil chord line and partially linearized boundary conditions imposed to compute the flow solution. The control points are moved from the airfoil surface to just above and below the airfoil chord line. The boundary condition is that the flow be in the direction tangent to the upper surface at the upper control point and the direction tangent to the lower surface at the lower control point (essentially, the upper and lower surface normal vectors are translated to points just above and below the chord line). The primary advantage of this method is the simplification of computing aerodynamic influence coefficients (AICs), since all of the angles between panels and control points are zero degrees except for the points in the very near field of the panels (for which the AICs are quite simple). This model works well for thin airfoils without much camber, but has significant problems around the leading edge of thicker airfoils and does very poorly on airfoils with large amounts of camber.

The failure of the partially linearized boundary condition model led to the development of a second model in which the control points are moved back out to the physical surface. In this model, the airfoil is covered with constant strength source panels, each of which has a control point at its center. A single point vortex is placed inside the airfoil to provide the circulation, and the Kutta condition is imposed to complete the system of equations. This model is analogous to a 3-d surface covered in quadrilateral source panels with a vortex lattice inside the wing. The results of this experiment are illustrated in Figure 5.1. The lift computed using this method matches almost exactly the lift computed using a 2-d full potential code for very low Mach number, but the pressure distribution is wrong in the immediate vicinity of the vortex (the vortex was placed on the airfoil mean line at mid-chord location). This experiment was encouraging for several reasons. The pressure distribution looks very good over most of the airfoil, and the lift is easily calculated by the Kutta-Joukowski theorem rather than requiring integration of pressures over the airfoil surface. Also, for a geometry with n panels, this method requires n(n+1) AIC calculations, as opposed to a source-doublet code that would require 2n*n AIC calculations. Placing several equal strength vortices inside the airfoil results in a pressure distribution with smaller discrepancies near the vortex location, but still yields a 'bumpy' distribution.

The final 2-d experiment was designed to keep most of the benefits of the source panel with discrete vortex method, while smoothing out the bumps in the pressure distribution. In this model the vortex is spread out as a set of constant strength vortex panels that follow the airfoil mean line. As before, there is only one vortex strength to solve for, it is just distributed along the airfoil mean line to provide a smooth velocity profile over the airfoil surface. The paneling is done so that there are an equal number of points on the upper and lower airfoil surfaces, and one mean surface panel corresponding to each pair of upper/lower surface panels. The number of AIC calculations for this case is now n(n+n/2), still less than the source-doublet case, and the system of equations is still (n+1) equations in (n+1) unknowns. This code has been demonstrated to compute lift

and pressure distributions for single-element and multi-element airfoil configurations that match within few percent of the results from the full potential code results.

The success of this final 2-d method led to the development of a new 3-d panel method that will be the base aerodynamic code used in this research. Constant strength quadrilateral source panels are placed on the surface of the wing, with vortex panels on the wing's mean surface. The geometry is illustrated in Figure 5.2 and Figure 5.3. The component of vorticity in the direction normal to the free-stream flow (the y direction) is set as constant from the leading edge to the trailing edge at any given span-wise station. The y-vorticity varies linearly in the y direction between each span-wise station, with the unknowns to solve for being the vorticity strengths at each of the span-wise stations. After specifying the y-vorticity distribution in terms of these unknowns, the component of vorticity in the x direction on the vortex panels and in the wake is computed according Helmholtz second vortex theorem. The boundary conditions for this formulation are tangent flow conditions at each surface control point and Kutta conditions at the trailing edge panel control points between each span-wise station. Since the y-vorticity strength is defined at each span-wise station, one additional boundary condition is required. Currently this condition is that the y-vorticity remain constant between the center-line and the first span-wise station. As in the 2-d case, there are several points that make the mean surface method an attractive alternative to a source-doublet surface panel code. Again, the number of AIC calculations required is considerably lower, and the lift can be computed using vortex theorems rather than integration of surface pressures. Another possible benefit is the improved condition of the system of equations to be solved for wings with very thin trailing edges. Doublet codes have historically shown conditioning problems when the upper and lower surface doublet panels get too close together, while the mean surface vortex has no upper or lower surface counterpart. The benefit of these features remains to be determined with further code testing.

The 3-d code currently gives very good results for some cases, but is not yet working satisfactorily for all general geometries. Pressure distributions and lift for unswept, untapered wings with various airfoil shapes agree within a couple of percent with results from A502, a relatively complex panel method used extensively in industry. A test case of an aspect ratio 6 NACA4412 wing was run on both the code described here and A502. The results were within 1% in lift, with excellent agreement in surface pressures. Furthermore, the research code took only 1 minute 49 seconds versus 5 minutes 17 seconds for A502.

The unswept, untapered case currently runs better than the general case for several reasons. Analytic formulas have been computed for the vortex panel AICs for rectangular panels only, which can be used to fit the simple wing geometry. More complex wing geometries, however, require more general trapezoidal vortex panel geometries, which, at this time, require numerical integration to compute the AICs. This numerical procedure requires considerably more time

than the analytic equations, and will be eliminated if possible in the future. The other problem with the current code lies in the final boundary condition discussed above. When wings are swept or tapered, the lift distribution changes and tends to be less 'flat' near the center. Forcing the vorticity to remain constant across that first span-wise panel, which really has no physical basis, forces the vorticity at the first inboard station to be either a little higher or a little lower than it really wants to be, which then cascades through the entire solution and results in a vorticity distribution that fluctuates up and down along the span rather than varying smoothly.

2.5.4 Research Activities Proposed for 1996-1997

During the coming year the models described above will be developed further and integrated to form a tool for wing optimization at the conceptual design level. The first task will be to complete the aerodynamic code, and validate it for a variety of general geometries. The final boundary condition must be corrected to alleviate the numerical error described above. Analytic formulas for the general vortex panel AICs will be sought, but in the mean time the numerical integration routine provides an opportunity to use the panel code to develop the rest of the design routines.

The sonic velocity prediction algorithm that was developed and tested on 2-d codes will now be implemented with the 3-d code. Comparison of the results with experimental data and/or Navier-Stokes simulations will be used to determine the flow conditions and geometries where this criteria is useful for determining maximum usable lift conditions. Finally, some criteria will be developed and the sonic velocity prediction routine will be added to the wing design method. The most important element of the design code to be developed in the coming year is the viscous model. First, an extensive literature search will be done to examine all of the current boundary layer computational methods, from 2-d integral methods to complete 3-d boundary layer simulations. The method used in this research must be relatively fast and simple, and it must account for the effects of sweep and abrupt changes in wing thickness and taper. The most likely result is a 2-d integral boundary layer method with some correction terms to account for transverse pressure gradients and cross-flow velocities in the inviscid solution. The viscous model will be coupled with the inviscid aerodynamic code to develop wing design tool that is sensitive to all of the available parameters that define the wing geometry. Comparisons will be made between this inviscid/viscous coupled code and full Navier-Stokes computations, as well as any available experimental results. Additionally, the design code will be used to look at some unconventional aircraft designs, such as the blended wing-body configuration, to see what insight this new tool gives to the aerodynamics and high lift characteristics of such designs.

2.5.5 *Research Activities Planned Beyond 1996-1997*

Once the 3-d aerodynamic model has been coupled with a viscous model to develop the desired wing analysis code, it will be ready for use with either an existing optimization package or possibly one to be developed in this project. The primary goal of this research is to design better wings faster, and a fast accurate analysis used in conjunction with numerical optimization methods is the key to accomplishing that goal. Several other useful design tools could also come as by-products of the development and testing of this high lift aerodynamic design package. A comparison of boundary layer properties on swept and unswept wings at similar conditions could be made to assess the applicability of using a 2-d boundary layer analysis on swept wings. This could be used to determine whether or to what extent the independence principle applies to the flow at conditions near maximum lift. Direct comparisons of velocity profiles, displacement thicknesses, momentum thicknesses and other boundary layer properties could be made between the 2-d wing, and along several cuts of a swept wing (such as normal to the sweep axis, aligned with the free-stream velocity or aligned with the inviscid streamlines). These comparisons could be used to assess the validity of using 2-d maximum lift data, and could lead to appropriate corrections to 2-d data based on properties such as inviscid pressure distributions or sweep and other geometric parameters. An attempt to build a Stratford like criteria for separation on a 3-d lifting surface could be made using data obtained from the research described above. In such a criteria, separation would be correlated with various properties of the flow such as pressure distribution, Reynolds number based on the distance the boundary layer has had to develop and the pressure gradient in the direction of local velocity. If such a criteria could be developed, it presents the best possibility for a fast but general method for preliminary assessment of maximum lift. Another key factor in modern wing design that would benefit from this improved high lift modeling is the prediction and alleviation of flutter. As the preliminary design code is refined to better assess wing high lift characteristics, a flutter analysis could be included. This would make the wing design algorithm substantially more valuable in the early stages of aircraft synthesis and optimization.

2.5.6 References

[5.1] Wakayama, S., "Lifting Surface Design Using Multidisciplinary Optimization," Ph.D. Dissertation, Stanford University Department of Aeronautics and Astronautics, Stanford, 1994.

[5.2] Jones, R.T., "Wing Theory," Princeton University Press, Princeton, NJ, 1990.

[5.3] Torenbeek, E., "Synthesis of Subsonic Airplane Design," Delft University Press, Delft, Holland, 1982.

[5.4] Furlong, G. Chester, and McHugh, James G., "A Summary and Analysis of Low Speed Longitudinal Characteristics of Swept Wings at High Reynolds Number," NACA Report 1339, 1957.

[5.5] Letko, William, and Goodman, Alex, "Preliminary Wind Tunnel Investigation at Low Speed of Stability and Control Characteristics of Swept Back Wings," NACA TN 1046,1946.

[5.6] Brune, G.W., and McMasters, J.H., "Computational Aerodynamics Applied to High-Lift Systems," Applied Computational Aerodynamics, Vol. 125 of Progress in Astronautics and Aeronautics, AIAA, Washington, D.C., 1990.

2.5.7 Figures

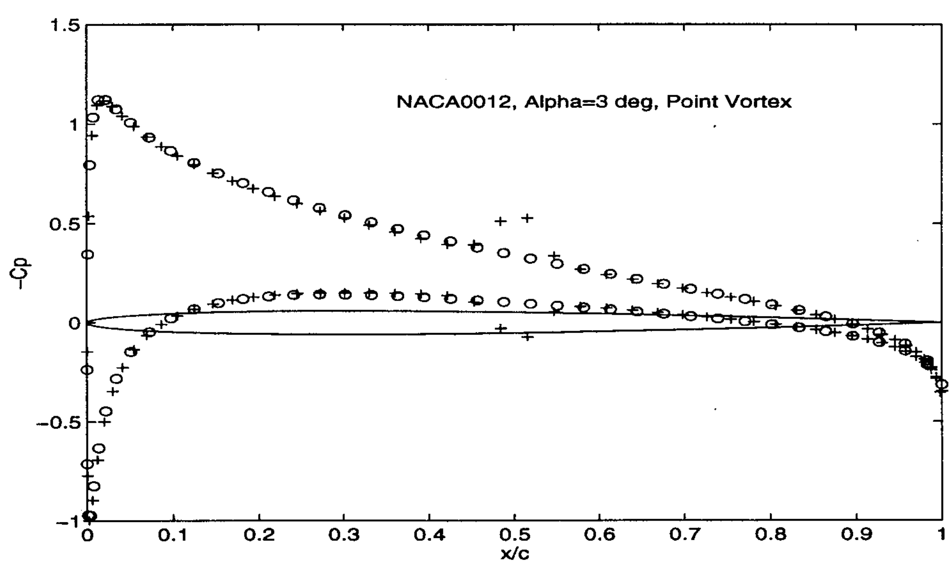

Figure 5.1 Pressure distribution of a NACA0012 airfoil at 3 degrees angle of attack. (+ = Source panel with discrete vortex code, o = Full potential code)

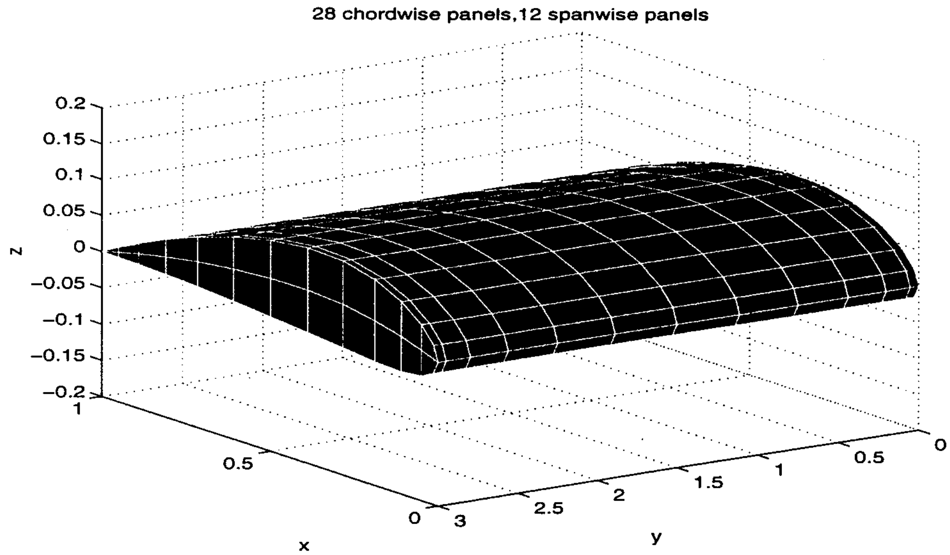

Figure 5.2 Source panel distribution for 3-d mean vortex panel code. NACA 4412, unswept, untapered wing; y=0 is the wing plane of symmetry.

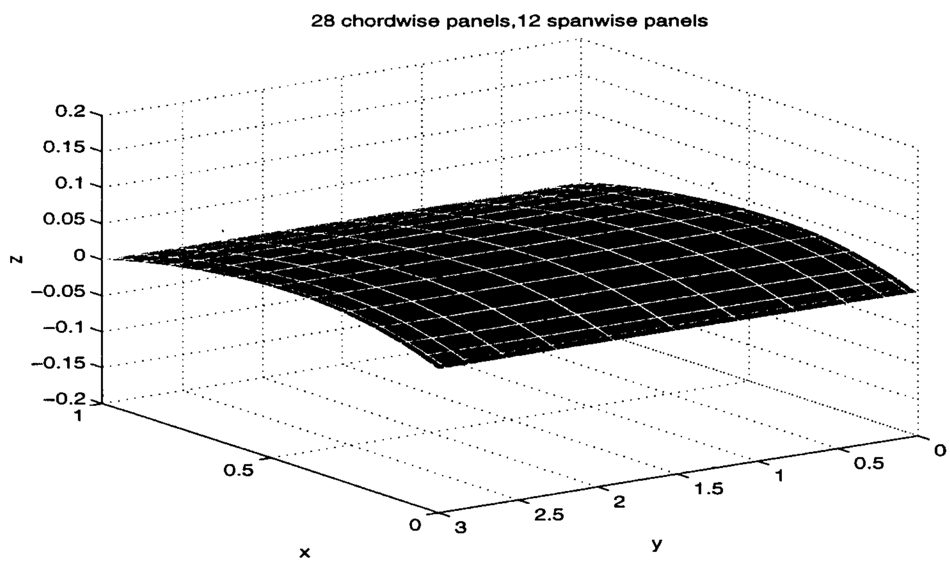

Figure 5.3 Vortex panel distribution for 3-d mean vortex panel code. NACA 4412, unswept, untapered wing.

2.6 PROJECT 6 - NEW METHODS FOR AERO-ACOUSTIC PREDICTIONS AT LOW SPEED

Research Participants: Prof. Sanjiva Lele, Graduate Student

Ames Technical Contact: Dr. N. Mansour

2.6.1 Introduction

The prediction and control (reduction) of aerodynamically generated noise is important to the design of quieter future aircraft. Airframe noise associated with high-lift devices in a landing configuration needs to be reduced in designing new subsonic airplanes. Wind noise impact of high-speed trains is also becoming increasingly important. High-speed trains travelling between city centers must pass through areas busy with urban communities and concerns about the community noise impact are rising. The common factor in these two examples is that the flow processes responsible for noise-generation occur at essentially incompressible conditions. Developing general computation based prediction methods for aerodynamically generated sound in the low Mach number regime is the focus of the proposed project.

Current prediction methods strongly rely upon the experimental database and use some form of Lighthill's acoustic analogy (Ref. [6.1]) for scaling the data. The predictive capability of such correlations is limited specially if significantly different new designs need to be examined. With the recent advances in computational algorithms and computer hardware (including parallel computing) a new generation of analysis tools for aero-acoustic prediction of low speed flows can be developed (Ref. [6.2]). Since these new tools are based on computations using the underlying physical principles, they offer the potential of being applied to configurations closer to the real hardware. The existing aeroacoustic theory, in order to remain tractable, requires many assumptions. While the theory has given useful scaling relationships, detailed solutions are available only for the simplest geometries and highly idealized flow situations.

The proposed project seeks to combine the theoretical ideas of asymptotic scaling and matching relevant to low Mach number aero-acoustics with computational solution methods. The development of such an approach should provide aero-acoustic predictions without the need for assuming a simplified geometry or idealized flow conditions.

2.6.2 Research Objective

Aeroacoustic theory and experiments indicate that in low Mach number sound generation problems, the flow region in the immediate vicinity of the sound sources' (unsteady flow, unsteady loads on surfaces, or moving surfaces) is adequately described as an incompressible flow. Away from such aero-acoustic source regions a sound field which is a superposition of the fields generated by all sources prevails. A sound wave is critically dependent on the medium/flow being compressible. Evidently, the exact governing equations expressing the conservation of mass, momentum and energy yield fundamentally different asymptotic balances in the two regions. The proposed research seeks to effi-

ciently exploit these asymptotic balances in a computational method for predicting low Mach number aerodynamically generated sound. The prediction method will be first tested on model problems for which theoretical predictions are available. The refined method will then be applied to more practical flow situations.

2.6.3 *Research activity proposed for 1996-1997*

We propose to carry out a program of research aimed at developing a new computational methodology for predicting the far-field sound generated by low Mach number aerodynamic flows. The new method is based on a composite computational solution which combines the near-incompressible near-field expansion (in the source region) with a wave-like far-field expansion of the unsteady aerodynamic flow. The principal investigator has carried out such expansions and different computational alternatives for combining them are available. In the first year of the project these alternatives will be implemented and tested on several model problems. Comparisons on these model problems, which allow theoretical solutions, will establish the most effective computational way of constructing a composite solution. The model problems of sound generated by transverse oscillations of a rigid cylinder, and the sound generated by vortex-shedding by flow passing over a fixed cylinder (relevant to landing gear noise) will be considered in this phase.

2.6.4 *Research activity planned for beyond 1996-1997*

In a second phase of this project the computational methods will be extended to consider the problem of sound generated by unsteady gusts interacting with an airfoil (leading edge noise). The acoustically compact limit will be considered first, and later extended to consider the diffraction/scattering effect of a non-compact airfoil chord. The computational method being developed aims to handle the compact and non-compact cases with the same numerical procedure. Achieving this is considered to be a significant milestone in the development of the new computational method. Current prediction methods require special Green's functions (or integral equation formulations) to describe the edge scattering/diffraction. The scattering from sharp edges (trailing edges of a wing, lateral edges of a flap, trailing edges of rotating blades) is currently believed to be a powerful noise source in low speed flows. A comprehensive prediction method for practical geometries which naturally handles this process does not exist. The research proposed should advance this field.

In the third phase of this project the computational method will be extended to consider three-dimensional aerodynamic flows. Three-dimensional computational models relevant to flap side-edge noise is one of the problems which can be studied. However, at the present stage it is clear that the simplest three-dimensional flow with practical noise implications should be treated first. If a broadband noise prediction is desired it will be necessary to incorporate the techniques of large-eddy simulation within the flow and noise prediction tool being developed under this project. A different, but equally useful direction,

would be consider inverse design problems with noise reduction aim while imposing aerodynamic performance and structural constraints. Both the leading edge noise, and the trailing edge noise are viable candidates for this purpose.

The computational method being developed can be easily adapted to noise prediction on more practical configurations. Such development, could occur in parallel with new efforts at NASA-Ames (involving participation from this project) in the third phase of this project. Our plan is to concentrate on the fundamental developments while keeping in mind the practical applications. It is expected that the first two phases of the project will take approximately two years to complete with some of the work proceeding in parallel. Once the capability for accurate predictions which include the edge scattering/diffraction is achieved the third phase of further developments of the method can begin.

2.6.5 References

[6.1] Lighthill, M. J. (1952) "On Sound Generated Aerodynamically: I. General Theory," Proceedings of The Royal Society, Ser. A, v. 211, p. 564-587.

[6.2] Crighton, D. G. (1988) "Goals for Computational Aeroacoustics," in Computational Acoustics: Algorithms and Applications, Elsevier Science Publishers B. V., IMACS, 1988.

3. PERSONNEL

The JIAA program described above will involve faculty, staff and graduate students as follows:

Faculty:

Prof. Brian Cantwell (PI), 15% academic year, 35% summer

Prof. Steve Rock, 5% academic year, 15% summer

Prof. Sanjiva Lele, 5% academic year, 15% summer

Prof. Ilan Kroo, 5% academic year, 15% summer

Staff:

Senior Research Associate: Dr. Rabi Mehta, 100% calendar year

Research Assistants: 5 Ph.D. students 50% AY/100% summer

Sci-Eng. Associate 10% calendar year

Throughout the conduct of this research and training activity close coordination will take place between the research personnel at Stanford and the research personnel and technical management staff at Ames.

4. FUNDING

The funding requested for the one-year period, Ref. October 1, 1996 to September 30, 1997 is given in the attached Estimated Cost Breakdown. As the University's contribution to the administration of the Institute, indirect costs on Professor Cantwell's administrative salary charges and administrative and secretarial support are waived.

ESTIMATED COST BREAKDOWN - JIAA 96-97 ADMINISTRATIVE BUDGET

GRANT NUMBER NCC 2-55
PROPOSAL NUMBER - AERO 97-04
DURATION - 12 MONTHS BEGINNING OCTOBER 1, 1996 TO SEPTEMBER 30, 1997

	ACADEMIC	SUMMER	CALENDAR	COST
A. SENIOR PERSONNEL				
B. Cantwell, Prof.	10.00%	20.00%		14,058
B. OTHER STAFF				
C. Edwards, Secy.			20.00%	6,677
TOTAL SALARIES and WAGES (A+B)				20,735
C. FRINGE BENEFITS (applied to TOTAL SALARIES AND WAGES)				
29.7% through 8/31/97, 25.5% through 8/31/98				6,086
TOTAL SALARIES, WAGES and FRINGE BENEFITS (A+B+C)				26,821
D. SUB-TOTAL DIRECT COSTS (A+B+C)				26,821
E. MODIFIED TOTAL DIRECT COSTS (D)				26,821
F. UNIVERSITY INDIRECT COSTS (Waived)				
0.00% through 8/31/96, 0.00% through 8/31/97				
G. ANNUAL AMOUNT REQUESTED (D+F)				26,821
TOTAL PROJECT COST				26,821

TOTAL ESTIMATED JIAA COST FOR 96-97	
Admin Budget	26,821
On-campus Budget	313,988
Off-campus Budget	150,089
ESTIMATED TOTAL JIAA 96-97 BUDGET	490,898

ESTIMATED COST BREAKDOWN - JIAA 96-97 ON-CAMPUS BUDGET

GRANT NUMBER NCC 2-55
PROPOSAL NUMBER - AERO 97-04
DURATION - 12 MONTHS BEGINNING OCTOBER 1, 1996 TO SEPTEMBER 30, 1997

	ACADEMIC	SUMMER	CALENDAR	COST
A. SENIOR PERSONNEL				
B. Cantwell, Prof.	5.00%	15.00%		8,435
S. Rock, Assoc. Prof.	5.00%	15.00%		8,000
I. Kroo, Assoc. Prof.	5.00%	15.00%		7,930
S. Lele, Assist. Prof.	5.00%	15.00%		7,100
B. STUDENTS				
STUDENT RESEARCH ASSISTANT	50.00%	50.00%		17,700
STUDENT RESEARCH ASSISTANT	50.00%	50.00%		17,700
STUDENT RESEARCH ASSISTANT	50.00%	50.00%		17,700
STUDENT RESEARCH ASSISTANT	50.00%	50.00%		17,700
STUDENT RESEARCH ASSISTANT	50.00%	50.00%		17,700
C. OTHER STAFF				
V. Matte, Sci.-Eng. Assoc.			10.00%	6,346
TOTAL SALARIES and WAGES (A+B+C)				126,311
D. FRINGE BENEFITS (applied to TOTAL SALARIES AND WAGES)				
29.7% through 8/31/97, 25.5% through 9/30/97				37,072
TOTAL SALARIES, WAGES and FRINGE BENEFITS (A+B+C+D)				163,383
E. OTHER COSTS				
Univ. services, communications, xerox, travel,pub.				8,000
F. COSTS NOT SUBJECT TO INDIRECT COSTS				42,500
Capital Equipment				
Project 1 - new model and control system modification - $8,500				
Project 2 - simulation display system-$3,000				
Project 3 - hot-wire and flow measuring system fabrication - $10,500				
Projects 1 and 3 - wind tunnel heat exchanger and freq. control system - $13,000				
Project 4 - pressure sensitive paint model fabrication measuring system - $7,500				
G. SUB-TOTAL DIRECT COSTS (A+B+C+D+E+F)				213,883
H. MODIFIED TOTAL DIRECT COSTS (G-F)				171,383
I. UNIVERSITY INDIRECT COSTS ON MTDC (H)				100,105
58.41% through 9/30/97				
J. ANNUAL AMOUNT REQUESTED (G+I)				313,988
TOTAL PROJECT COST				313,988

ESTIMATED COST BREAKDOWN - JIAA 95-96 OFF-CAMPUS BUDGET

GRANT NUMBER NCC 2-55
PROPOSAL NUMBER - AERO 97-04
DURATION - 12 MONTHS BEGINNING OCTOBER 1, 1996 TO SEPTEMBER 30, 1997

	ACADEMIC	SUMMER	CALENDAR	COST
A. SENIOR PERSONNEL				
B. Cantwell, Prof.				
B. OTHER STAFF				
R. Mehta, Senior Res. Assoc.			100.00%	85,513
TOTAL SALARIES and WAGES (A+B)				85,513
C. FRINGE BENEFITS (applied to TOTAL SALARIES AND WAGES)				
29.7% through 8/31/97, 25.5% through 8/31/98				25,090
TOTAL SALARIES, WAGES and FRINGE BENEFITS (A+B+C)				110,603
D. SUB-TOTAL DIRECT COSTS (A+B+C)				110,603
E. MODIFIED TOTAL DIRECT COSTS (D)				110,603
F. UNIVERSITY INDIRECT COSTS ON MTDC (H)				39,485
35.7% through 9/30/97				
G. ANNUAL AMOUNT REQUESTED (F)				150,089
TOTAL PROJECT COST				150,089